N69-37572
NASA CR-102247
01977-6025-R0-00

MISSION ORIENTED STUDY
OF
ADVANCED NUCLEAR SYSTEM PARAMETERS

PHASE VI FINAL REPORT

VOLUME I

SUMMARY TECHNICAL REPORT

JUNE 1968

for

George C. Marshall Space Flight Center
National Aeronautics and Space Administration

TRW
SYSTEMS GROUP

Volume I SUMMARY TECHNICAL REPORT

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FOREWORD

This volume, which is the first of a set of two volumes, summarizes the study tasks, analyses, and results of the Mission Oriented Study of Advanced Nuclear System Parameters, performed under Contract NAS8-5371, for George C. Marshall Space Flight Center, Huntsville, Alabama. This work was performed during the period from July 1967 to June 1968 and covers Phase VI of the subject contract.

The final report has been organized into a set of two separate volumes on the basis of contractual requirements. The volumes in this set are:

01977-6025-R0-000	Volume I	Summary Technical Report
01977-6026-R0-000	Volume II	Technical Report

Volume I summarizes and Volume II presents the details of the basic study guidelines and assumptions, the analysis approach, the analytic techniques developed, the analyses performed, the results obtained, and an evaluation of these results together with specific conclusions and recommendations. Also included in these two volumes are discussions of those areas of research and technology in which further effort would be desirable based on the results of the study.

This study was managed and principally performed by personnel in the Analytical Research Operations of the Systems Laboratories of TRW Systems. The principal contributors to this study were Messrs. G. M. Callies, A. R. Chovit, R. S. Schussler, and L. D. Simmons.

ABSTRACT

A summary is given of the study approach and basic guidelines and assumptions which were used in a series of analyses of manned Mars lander and manned Mars and Venus orbital capture (no manned lander) missions. Analyses were performed for Mars missions employing opposition class, Venus swingby and conjunction class trajectories for launch opportunities from 1980 through 1993; the Venus missions employed inferior conjunction class trajectories for launch opportunities from 1980 through 1985. The investigations included comparative analyses of vehicles using cryogenic chemical, liquid storable, and nuclear rocket propulsion systems; nuclear rocket thrust levels of 75,000, 100,000 and 200,000 lb were analyzed. Both circular and elliptic Mars parking orbits were investigated and an analysis of Earth and Mars launch window requirements was made for selected missions. A summary of the analyses and results is presented as well as an evaluation and recommendations based on the results.

CONTENTS

ILLUSTRATIONS

TABLES

I. INTRODUCTION

This summary volume presents in condensed form the results of Phase VI of the Mission Oriented Study of Advanced Nuclear System Parameters performed by TRW Systems for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration. The report includes, in addition to the results of the analyses, the objectives of this phase of the study and the guidelines and constraints used in the analyses.

Phase VI of the study can be clearly divided into two successive parts because of the redirection of the work at approximately the mid-point of the contract period. The redirection was prompted by renewed interest in the development of a nuclear engine in the 75, 000- to 10, 000-lb thrust range rather than the previously assumed 200, 000-lb thrust level. This change, together with the allocation of other study objectives to the first or second parts of the period, led to the division of tasks indicated in Table I-1.

Table I-1. Study Tasks by Phase

Phase A 200, 000-lb Thrust Nuclear and Chemical Engines	Phase B 75, 000- and 100, 000-lb Thrust Nuclear Engines
1. Optimized Venus swingby missions, 1980-1993	1. Optimized Venus swingby and opposition class missions, 1984-1993
2. Optimized opposition class missions, 1980-1993	2. Optimized Mars orbital capture mission, 1984 opposition class, circular and elliptic Mars parking orbits
3. Optimized conjunction class mission, 1983	3. Same as (2) above, except with nuclear engine aftercooling
4. Optimized orbital capture missions* to Mars and Venus 1980-1985	4. Effects of providing launch windows at Earth and Mars

*Orbital capture mission refers to a mission in which unmanned probes are deployed to the surface and not recovered. Unless otherwise specified, all missions are manned Mars lander missions.

The work reported here was a continuation of that performed under the earlier phases (I through V) of this study and made use of the techniques and computer programs developed under those phases. References 1 through 6 contain the details of these techniques and programs as well as the results of the earlier analyses. In addition, Volume II of this report includes a brief description of the SWOP Computer Program used in the analyses.

As in the previous analyses, the criterion of optimization was minimum initial vehicle weight prior to launch from Earth orbit. Trajectory types, vehicle configuration, and engine clustering were varied to obtain the minimum weight for each type of mission investigated.

II. MISSION AND VEHICLE PERFORMANCE PARAMETERS

The types of missions investigated in the present study were the following:

Mars Missions

- Opposition class
- Venus swingby
- Conjunction class
- Orbital capture (with unmanned surface probes)

Venus Mission

- Orbital capture

Definitions of these mission types will be found in the earlier studies (References 1 and 6) and in Volume II of this report. The orbital capture missions differ from lander missions only in that there is no manned lander and the surface probes are not recovered; the stay time at the planet in both cases is nominally 30 days (except for conjunction class missions with their optimized, extended stay times).

Trajectory types were selected on the basis of previous studies (Phase III), where it was shown that the IIB opposition class trajectory generally results in the minimum weight for all opportunities. This type of trajectory (i. e., outbound leg less than 180° heliocentric transfer angle, inbound leg greater than 180°) was used for all opposition class missions except in those cases where the launch azimuth constraints were violated and the type IB trajectory was used. The trajectory types 3 and 5 were used for the swingby leg of swingby missions.

The 72° to 114° launch azimuth constraints for ETR launches were used in the analyses, limiting the departure asymptote declinations to -36° to $+36^{\circ}$ if plane changes are to be avoided.

The parameters assumed for the different propulsion systems considered are shown in Table II-1, with engine weights and thrust levels for clusters of nuclear engines taken as direct multiples of the corresponding values.

Table II-1. Propulsion System Performance Parameters

Type	Specific Impulse (sec)	Thrust (lb)	Engine Weight (lb)
Nuclear	850	200,000	30,750
	850	100,000	20,000
	850	75,000	18,000
Cryogenic	460		
Storable	380		

Impulsive velocities were computed on the basis of injection into interplanetary orbit from a 500-km circular Earth parking orbit and braking into a 600-km circular parking orbit at the target planet. For elliptic orbits, the periapsis was taken as 600 km and the ratio of periapsis to apoapsis radius as six. Allowance was made for gravity losses resulting from finite thrusting in a gravity field.

For chemical cryogenic propulsion, a tanking mode was assumed. This means that all propellant tanks are assumed to be full at the time of launch from Earth orbit, having been filled from a vehicle while in Earth parking orbit. Maximum capacity for a given tank is determined by the Saturn V payload weight limitation or overall vehicle length limitation; a module may be launched from Earth either empty or partially full.

In the case of nuclear propulsion, a connecting mode was assumed. In this mode, all modules are placed into orbit fully loaded with propellant. Additional propellant, where required, is provided by adding a propellant module (tank only, without engine). A typical vehicle configuration for the connecting mode is shown in Figure II-1, where the propulsion modules are shown with engines attached and the propellant modules without engines. In this case, the leave Earth stage consists of a cluster of three propulsion modules (Tier 1). If more propellant is required, a single propellant module or three propellant modules are added (Tier 2A or 2B). If this amount is still insufficient, a fourth propellant module is added as Tier 3. The same procedure is used in configuring the arrive Mars and leave Mars stages, although normally only one propulsion module is required for these stages when the 200,000-lb thrust engine is used. In the Phase B analyses of 75,000-and 100,000-lb thrust nuclear engines, clustering was

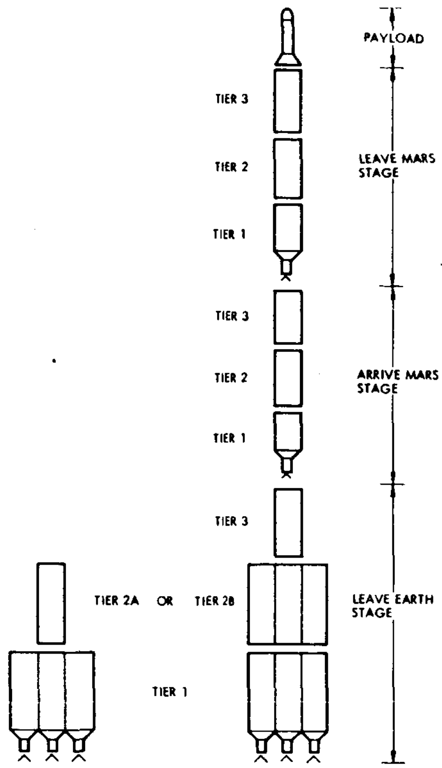

Figure II-1. Typical Connecting Mode Vehicle Configuration

considered for the first two main propulsion stages. The optimum number of engines in a cluster was taken to be that number which yielded the minimum weight vehicle in Earth orbit without exceeding a maximum nuclear engine firing time constraint of 2700 sec.

For all vehicles, allowance is made for attitude control, midcourse correction stages, a planetary orbit adjustment maneuver, and an arrive Earth retro stage where required. These are all assumed to use storable chemical propulsion.

In the case of chemical propulsive stages, micrometeroid protection and insulation are allowed for. For nuclear stages a combined micrometeoroid and heat protection shield is assumed to reduce boiloff during outbound transit and planetary stopover periods. This shield is jettisoned just prior to the arrive planet and depart planet propulsive maneuvers.

A block weight is assigned to each nuclear stage to allow for radar, docking and interstage structure, attachment members, and the separating mechanism. The required characteristic velocity for each stage was increased by 0.75% to provide a propellant reserve.

The scaling laws used to determine the stage weights are given in Volume II of this report and are those provided by MSFC for this study. Payload and expendable weights were also provided by MSFC and are given in Table II-2. The augmented crew size and payload weights for the conjunction class missions are dictated by the long stay time at Mars (over a year) for this type of mission.

Table II-2. Payloads and Expendable Weights

Payload	Orbital Capture	Stopover Lander	Conjuction Class
Crew size (men)	6	8	12
Earth return module (lb)	12,600	12,800	16,120
Mission module (lb)	180,000	180,000	110,000
Solar flare shield (lb)	14,000	14,000	20,000
Mars excursion module (lb)	None	100,000	150,000
Mars orbit return weight (lb)	None	1,500	3,000
Drop weight for capture missions (lb)	35,000 (Mars) 20,000 (Venus)		
Life support expendables (lb/day)	30	35	50

The scaling laws used to size the aerodynamic braking systems were developed during Phase III of the study and are described in Reference 1. The values used are also presented in Volume II of this report, as are the relationships used to compute the insulation and propellant boiloff weights.

The code used to designate the various vehicle configurations is the same as that used in previous reports. The symbols shown in Table II-3 are combined into a code of which each element indicates the means of effecting the energy change for each major acceleration/deceleration maneuver, as follows:

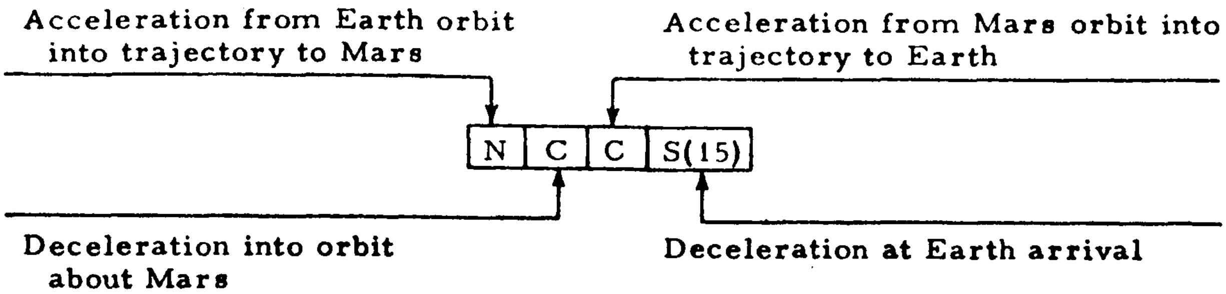

Table II-3. Vehicle Configuration Symbols

Symbol	Means of Energy Change
N	Nuclear engine
N_J	Aftercooled nuclear engine; jettison propellant tankage (100, 000-lb thrust)
N_{NJ}	Aftercooled nuclear engine; retain propellant tankage (100, 000-lb thrust)
C	Chemical cryogenic
S	Storable chemical propulsion
A	Aerodynamic braking
S(15)	Storable chemical retropropulsion to 15 km/sec followed by aerodynamic braking

III. PHASE A MISSION ANALYSES

As noted in the introduction, the Phase A analyses consisted of computing optimum Mars lander missions for the 1980 to 1993 launch opportunities, considering Venus swingby, opposition class, and conjunction class missions, plus analyses of orbital capture missions to Mars and Venus for their corresponding opportunities in the 1980 to 1985 period. All of these analyses included both chemical cryogenic propulsion systems and the 200,000-lb thrust nuclear engine.

The permissible departure asymptote declinations resulting from the ETR launch azimuth constraints fall in the range -36° to $+36^{\circ}$. Each optimized mission was examined to determine whether this constraint was violated, and in those cases where the minimum weight trajectories led to departure asymptotes outside this range, the opposite type of outbound leg (i.e., type I instead of type II) was used. Table III-1 lists those missions where this change in trajectory was required.

Table III-1. Missions Violating the Launch Azimuth Constraint

Mission Type	Trajectory Type	Launch Opportunity	Declination (deg)
Opposition class	IIB	1980	+39.9 to +55.3
		1984	-35.9 to -37.0
		1986	-50.1 to -51.5
		1990	+37.6 to +41.6
		1993	+43.7 to +49.9
Venus swingby	II5	1984	-36.0
	5A, 5B	1990	-64.0

In all cases, the vehicle configuration for the nuclear propelled vehicle was basically that illustrated in Figure II-1, with three nuclear engines and their associated propulsion modules used for Earth depart and one such module for planet arrive and planet depart maneuvers.

This is referred to as the 3-1-1 configuration. Propellant modules were added as necessary to this basic configuration. A maximum firing time constraint of 2700 sec was applied to the nuclear engines, but was not exceeded in any of the missions analyzed.

The matrix of mission, trajectory, and vehicle types analyzed for the lander mission is given in Table III-2.

Table III-2. Mars Lander Mission Matrix

Mission Type	Launch Opportunity	Trajectory Type	Vehicle Mode
Venus swingby	1980-1993	Swingby leg - 3 and 5 Direct leg - I, II, A and B	NNNA and CCCA
Opposition class	1980-1993	IIB	NNNA, NNNS(15), CCCA, and CCCS(15)
Conjunction class	1983	IA	NNSA and CCSA

Complete data for each combination analyzed are presented in tabular form in Volume II, including initial vehicle weight, total trip time, Earth arrival speed, and Julian dates of Earth depart, planet depart, and Earth arrival. These data are presented here in the form of summary charts indicating initial vehicle weight in Earth orbit for selected combinations of trajectory and vehicle types for each opportunity.

VENUS SWINGBY MISSIONS

Figure III-1 presents these results for manned lander missions to Mars over the 1980 to 1993 period where either the outbound or the inbound leg is constrained to pass in the vicinity of Venus performing a hyperbolic turn about the planet. Arabic numerals are used to designate the swingby leg, in accordance with the scheme developed by Ross and Gillespie (Reference 7). The direct leg is designated by the previously mentioned symbols.

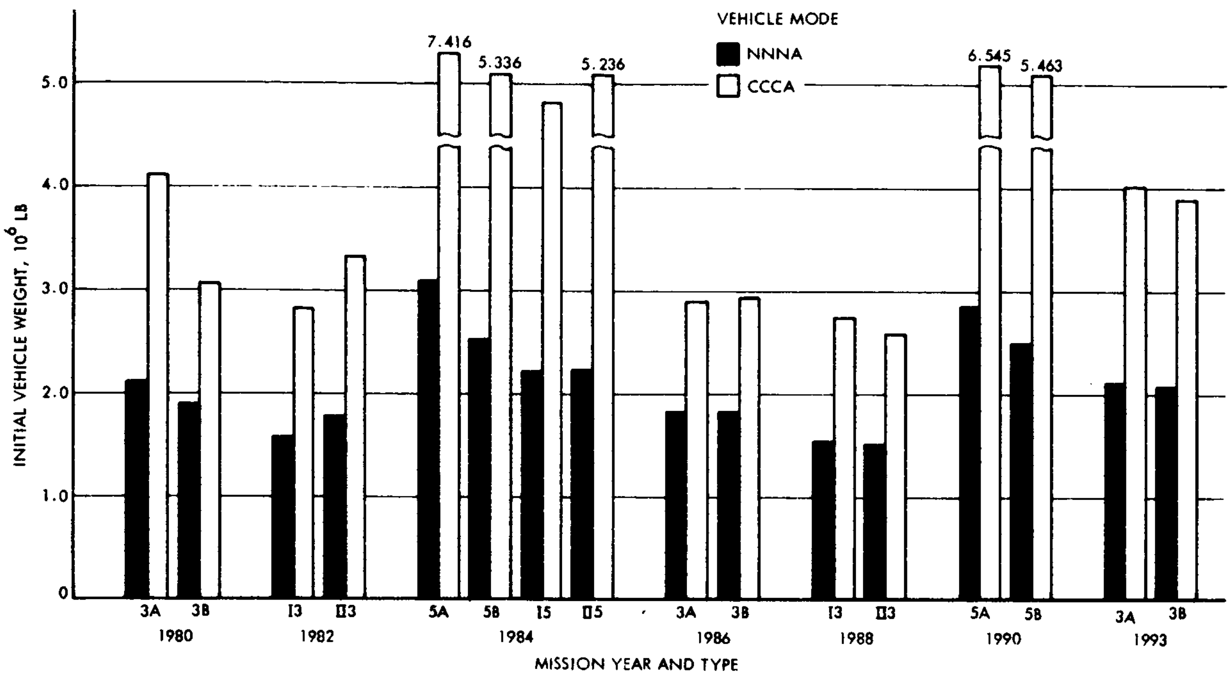

Figure III-1. Venus Swingby Lander Mission

For each opportunity, all feasible swingby trajectories were analyzed for the NNNA and CCCA vehicle modes. It will be noted from the figure that in all opportunity years except 1984, swingby missions must use either inbound or outbound swingbys, while in 1984 both inbound and outbound swingbys are possible. As indicated in Table III-1, both 1990 swingby missions violate the launch azimuth constraint and the 1984/II5 mission is on the borderline in this respect. The figure also shows that except in 1986 and 1988, the minimum weight vehicle results when the swingby leg is combined with the long direct leg (I or B).

The usual large differences between chemical cryogenic and nuclear propulsion appear in this analysis, averaging out to a factor of approximately two. The minimum weight vehicles for this cycle of launch opportunities range from about 1.53 to 2.22M lb, excluding the unacceptable 1990 missions.

OPPOSITION CLASS MISSIONS

For this class of mission, four vehicle types were considered: all nuclear propulsion with and without aerodynamic braking at Earth arrival, and all chemical cryogenic propulsion with the same two options (in both cases the retropropulsion at Earth is provided by a storable chemical stage). The results of this analysis are summarized in Figure III-2. The trajectory types indicated are those resulting in minimum weight vehicles while observing the launch azimuth constraint; in general, the IIB trajectory yields the lowest weights but has been changed to IB in the years indicated because of the launch azimuth constraint.

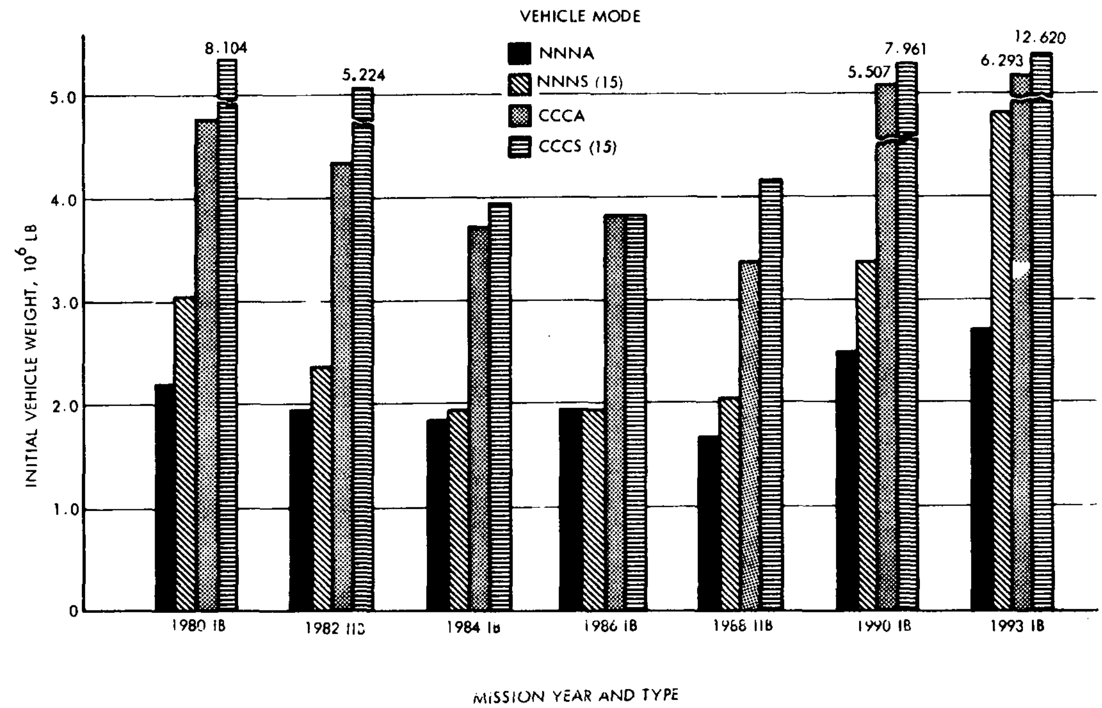

Figure III-2. Mars Opposition Class Lander Mission

The results are typical of those obtained from similar previous analysis, with nuclear propulsion affording a weight reduction by about 50% and all aerodynamic braking at Earth arrival reducing weight requirements by a third to a half during the unfavorable opportunities in the synodic cycle (as compared to aerodynamic braking limited to velocities of 15 km/sec).

CONJUNCTION CLASS MISSIONS

A 1983 conjunction class mission was analyzed for comparison purposes. A Type IA trajectory was used, and both NNSA and CCSA vehicle configurations were considered. The optimum stopover time for this mission is 416 days, and total trip time is 956 days. The resulting initial vehicle weights are 1.638 and 2.496M lb for the nuclear and chemical cryogenic systems, respectively (in both cases, the Mars depart stage was assumed to be a liquid storable system, because of propellant boiloff considerations of cryogenic propellants over the long Martian stopover period).

SUMMARY OF PHASE A LANDER MISSION RESULTS

Figure III-3 presents a summary comparison of Venus swingby and opposition class missions for the period considered, using the NNNA vehicle configuration. The 1983 conjunction class mission (using the NNSA vehicle) is included for comparison. The minimum weight mission has been selected in each case, with the exception that for the opposition class missions, the launch azimuth constraint has been applied even where it leads to a higher initial vehicle weight.

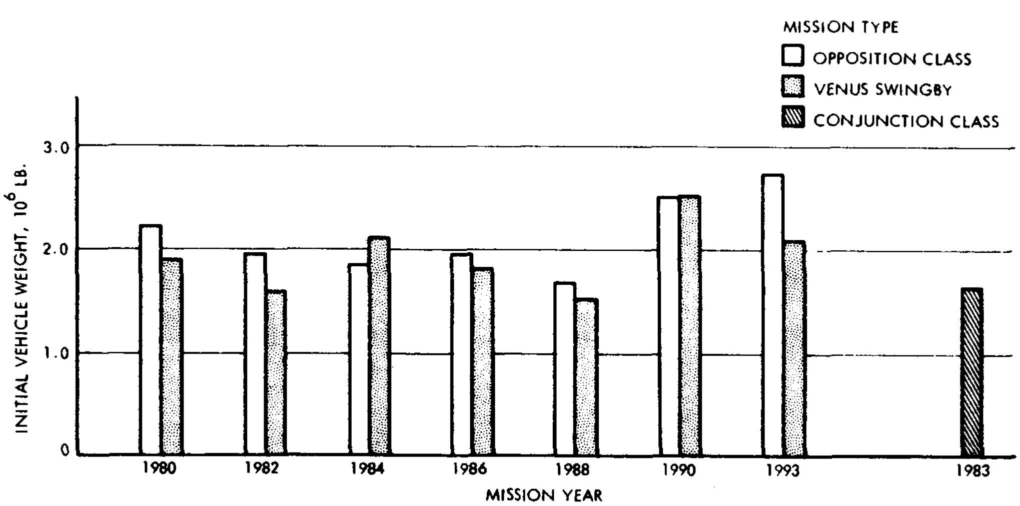

Figure III-3. Lander Mission Mode Comparison — NNNA Vehicle Configuration

In all years except 1984 and 1990 the Venus swingby mission requires a lower weight vehicle than does the opposition class mission (and in any case the 1990 swingby mission violates the launch azimuth constraint). The range of minimum vehicle weights is from 1.53 to 2.51M lb. The 1983 conjunction class mission shows a lower vehicle weight than any of the others except the 1982 and 1988 swingby missions.

ORBITAL CAPTURE MISSIONS

The final task performed under Phase A of the study was the analysis of manned missions to Mars and Venus which do not provide for a manned, recoverable lander but instead deploy unmanned, instrumented probes to the planet surface. The same vehicle and trajectory types and launch azimuth and firing time constraints were considered as in the lander mission analyses. One difference was that the stopover time at the target planet (the period during which the manned vehicle orbits the planet and makes observations) was varied from 20 to 40 days instead of being fixed at 30 days.

The matrix of mission and trajectory types and vehicle configurations analyzed is given in Table III-3. It should be noted that this part of the study was limited to opportunities in the 1980 to 1985 period.

Table III-3. Orbital Capture Mission Matrix

Target Planet	Mission Type	Launch Opportunity	Trajectory Type	Vehicle Mode
Mars	Opposition class	1980-1984	IIB	NNNA, NNSA, NNNS(15), CCCA, and CCCS(15)
	Venus swingby	1980-1984	Swingby-3 and 5 Direct Leg-I, II, A and B	NNNA, NNSA, NCCA, CCCA, and CCSA
Venus	Inferior conjunction	1980-1985	Outbound-I and II Inbound-A and B	NNNA, NNSA, NCCA, NSSA, CCCA, and CCSA

The effect of varying planetary stopover time from 20 - 30 to 40 days was investigated for each type of mission identified in the matrix of Table III-3. For opposition class missions falling within the period considered (1980, 1982, and 1984), the initial vehicle weight was found to increase by 6 to 9% for each 10-day increase in stopover time. In the case of Venus swingby missions, the same weight variation was found for the 1984 opportunity, but for the 1980 and 1982 opportunities, the change in vehicle weight was less than 1% for each 10-day increment in stopover time. For missions to Venus, each 10-day increase in stopover time resulted in a 2 to 4% increase in weight in 1980 and 1982, but no significant change in 1983 and 1985. Complete trajectory and weight data for all missions analyzed are given in Volume II of this report.

Initial vehicle weights for orbital capture missions to Mars using opposition class trajectories and various vehicle configurations are shown in Figure III-4. The results are the same as those for manned lander missions in that constraining Earth aerodynamic braking capability to 15 km/sec increases vehicle weight by a factor of 1.5 to 2, and nuclear propelled vehicles weigh approximately half as much as chemically propelled vehicles.

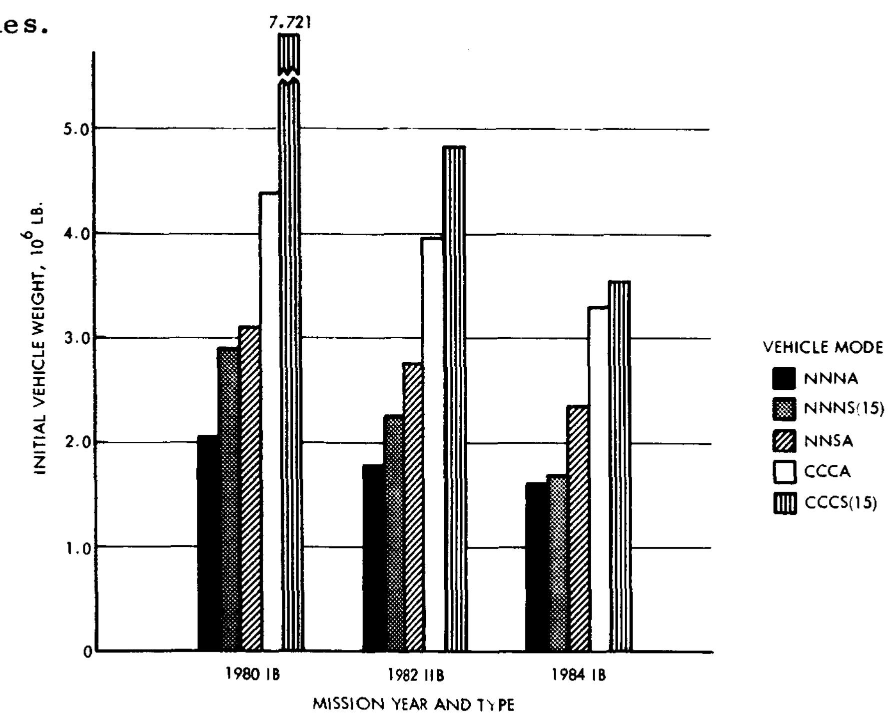

Figure III-4. Mars Orbital Capture Mission - Opposition Class, 30-Day Stopover Period

Comparable results for Venus swingby missions to Mars are shown in Figure III-5. In all cases, the minimum weight trajectories are those combining a long direct leg (I or B) with the swingby leg. Minimum vehicle weights range from about 1.44 to 2.01M lb.

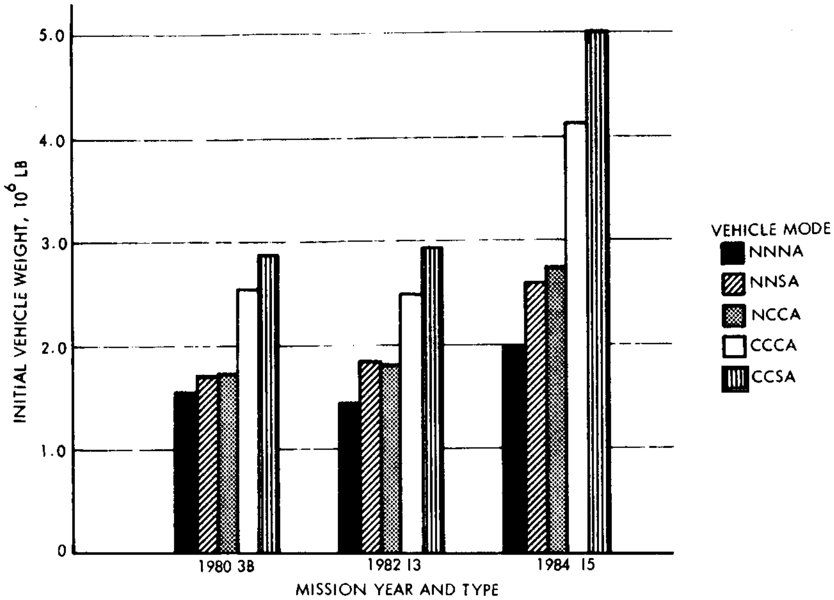

Figure III-5. Mars Orbital Capture Mission-Venus Swingby 30-Day Stopover Period

Comparisons of the same vehicle types for orbital capture missions to Venus are shown in Figure III-6, in all cases using a Type IIB inferior conjunction trajectory. The weight variations over the period are relatively slight; the NNNA vehicle provides the lowest weights, in the range of 1.45 to 1.65M lb.

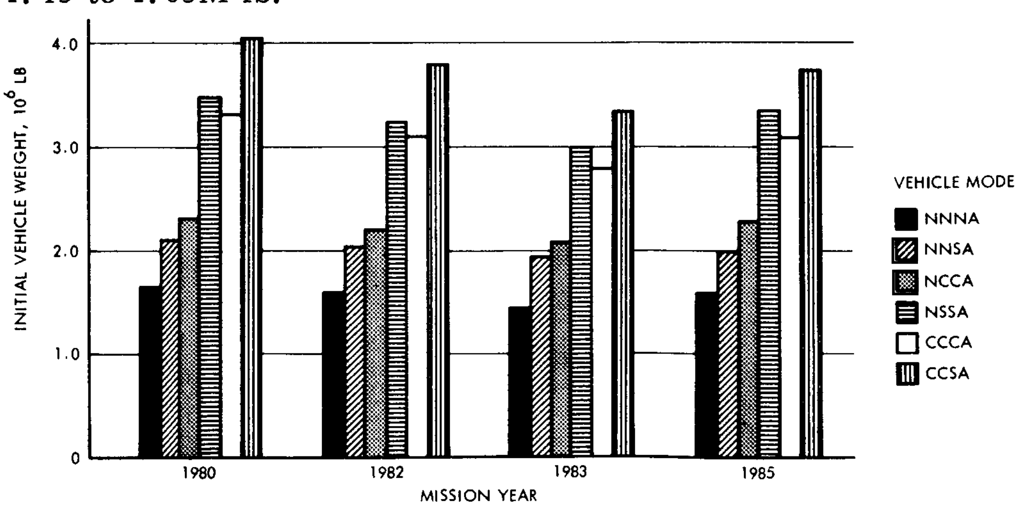

Figure III-6. Venus Orbital Capture Mission Type IIB 30-Day Stopover Period

From the above data, the minimum-weight missions have been selected and are compared in Figure III-7. The results are identical to those for the Mars lander missions in that the Venus swingby trajectories yield lower vehicle weights than do the opposition class trajectories for all years except 1984. The figure also indicates that the minimum weight vehicles for orbital capture missions to either Mars or Venus during this period fall in the relatively narrow range of 1.44 to 1.65M lb. Comparable lander missions (to Mars only) require approximately 170,000 to 350,000 lb additional weight.

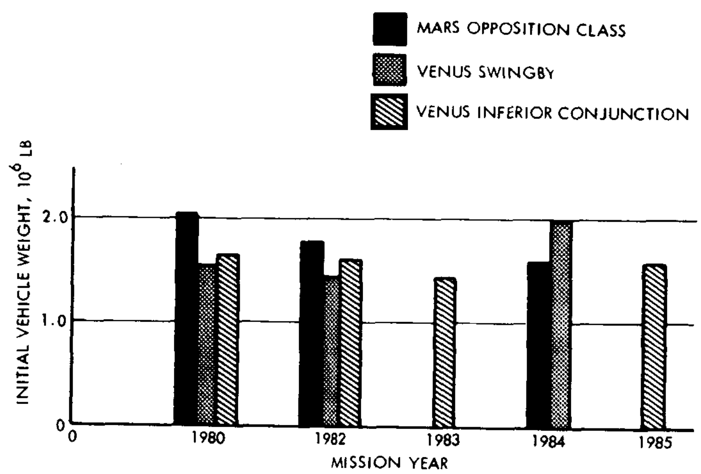

Figure III-7. Orbital Capture Mission Mode Comparison NNNA Vehicle Configuration

IV. PHASE B MISSION ANALYSES

As indicated earlier, the second part of the Phase VI study was devoted primarily to evaluating missions in which the nuclear engine to be used was much smaller than the 200,000-lb thrust engine used in previous analyses. Values of both 75,000 and 100,000 lb were to be used, with optimum clustering to produce the required velocity changes. In addition, certain of the missions analyzed were to be considered with elliptic as well as circular parking orbits at Mars. The possibility of aftercooling was also to be investigated: a nuclear engine is cooled after being shut down by means of an additional flow of hydrogen through the reactor and then restarted for a later propulsive maneuver rather than jettisoned. The tradeoff is one between the weight of the hydrogen required for aftercooling and the weight of the added nuclear engine if aftercooling is not used.

Finally, an evaluation was to be made of the effect on total initial vehicle weight in Earth orbit of providing launch windows at Earth and at Mars. Previous analyses in this study were based on the assumption of on-time launches at both points.

To permit useful comparisons with the earlier studies while restricting the scope of analyses to the evaluation of the parameters under investigation, only Mars missions with 30-day stopover periods were considered. The same maximum firing time constraint of 2700 sec was used, as was the ETR launch azimuth constraint limiting departure asymptote declinations to the range of -36.6 to + 36.6^{o} and the parking orbit inclination to 28.4 to 36.6^{o}. Scaling laws and system weights were unchanged except for the required new scaling laws used in sizing stages for the smaller nuclear engines. These were derived for each of the engine clustering arrangements considered and are given in Appendix A of Volume II. Nuclear engine weights and performance parameters for these engines were given in Table II-2.

LANDER MISSION ANALYSIS

The objective of this task was to evaluate the effects of using the smaller nuclear engines. Launch opportunities in the 1984 to 1993 period were investigated, considering only the NNNA vehicle and circular parking orbits at Mars. Table IV-1 lists the missions analyzed in this task; they are the missions determined in Phase A to require the minimum weight vehicle within the specified launch azimuth constraints.

Table IV-1. Mars Lander Mission Matrix

Launch Opportunity	Mission Type	Trajectory Type
1984	Opposition class	IIB
1986	Outbound Venus swingby	3A
1988	Inbound Venus swingby	II3
1990	Opposition class	IB
1993	Outbound Venus swingby	3A

The first step in the analysis was to determine the optimum engine clustering arrangement for each of the above missions. In all cases, a single nuclear engine was sufficient for the depart Mars stage; the optimization problem was thus reduced to determining the optimum clustering for the leave Earth and arrive Mars stages. The SWOP computer program was used to find the minimum vehicle weight for the various possible clusterings of leave Earth and arrive Mars engines. This process was carried out for each of the launch opportunities. The results, presented in detail in Volume II, are summarized in Table IV-2; the table also includes the corresponding values for missions using the 200, 000-lb thrust nuclear engine.

The general conclusions that can be drawn from the above data are as follows:

- The 100, 000-lb thrust engine provides a total vehicle weight from 3 to 10% less than that resulting from use of the 75, 000-lb thrust engine, and use of the 200, 000-lb thrust engine affords a further reduction of from 6 to 13%. As compared to the 75, 000-lb thrust engine, the 200, 000-lb thrust engine affords a reduction of from 9 to 18%.

- With respect to the number of engines required, use of the 100, 000-lb thrust engine reduces the number required by at least one and, in two cases, two as compared to the 75, 000-lb thrust engine. As many as four fewer engines are required when the 200, 000-lb thrust engine is used.

Table IV-2. Mars Lander Missions Clustering and Weights

Mission	75,000-lb Thrust			100,000-lb Thrust			200,000-lb Thrust		
	Vehicle Weight (10^6 lb)	Configuration	Number Modules	Vehicle Weight (10^6 lb)	Configuration	Number Modules	Vehicle Weight (10^6 lb)	Configuration	Number Modules
Opposition 1984 IIB	2.129	4-2-1	8	1.911	3-1-*1	7	1.798	2-*1-*1	8
Venus Swingby 1986 3A	2.134	4-2-1	8	1.936	3-*1-*1	8	1.790	2-*1-*1	8
Venus Swingby 1988 II3	1.599	3-1-1	6	1.545	2-*1-1	7	1.455	**1-1-1	6
Opposition 1990 IB	-	-	-	2.423	4-*1-1	9	2.208	**2-1-1	9
Venus Swingby 1993 3A	2.507	5-2-1	9	2.355	4-2-1	8	2.046	2-*1-*1	8

*Tier 2A required

**Tiers 2 and 3 required

- On the basis of the "in-line" concept of adding propellant modules illustrated in Figure II-1, an increase in the engine thrust level may or may not reduce the number of required modules (a module is a propellant module, a propulsion module with engine, or the payload plus secondary systems.) Use of the "in-line" concept often results in an inefficient application of the connecting mode concept.

ORBITAL CAPTURE MISSION ANALYSIS

The procedures, performance parameters, and constraints used in the analysis of Mars orbital capture missions were the same as in the lander mission case, with two exceptions: 1) both elliptical and circular parking orbits at Mars were considered; 2) both aftercooled and non-aftercooled nuclear engines were considered for the maneuvers at Mars. Only the 1984 IIB opposition class mission was investigated.

The procedure for analyzing the elliptic parking orbit cases is described in detail in Volume II; in general, the orientation of the ellipse and the location on the ellipse of the impulsive maneuvers were optimized to yield a minimum vehicle gross weight. As noted earlier, the periapsis of the ellipse was taken as 600 km in all cases and the apoapsis such as to make the ratio of periapsis to apoapsis equal to six. In a more detailed analysis the apoapsis altitude would be optimized for each case, but the complexity of such an optimization was not warranted in this preliminary investigation of the effects of using elliptic parking orbits.

Two different staging arrangements were investigated for the use of aftercooled engines. In one, designated NN_JNA, the arrive Mars stage is provided with a propellant module containing the propellant for braking into Mars orbit and for aftercooling the engine. This module is then jettisoned, and a separate propellant module used for the depart Mars manuever. In the other mode, designated $NN_{NJ}NA$, a single propulsion module contains all propellant required for braking into orbit, after-cooling, and departing Mars. The equation used to compute the required amount of aftercooling propellant was provided by MSFC and is given in Volume II.

The results of this analysis are presented in detail in Volume II and are summarized in Table IV-3. The NNNA vehicle was analyzed with the 75,000-and 200,000-lb thrust nuclear engines, while for the 100,000-lb thrust engine, the same vehicle was considered without aftercooling and with each aftercooling mode. The NNSA vehicle configuration was also included for comparison. The major conclusions to be drawn from this table can be summarized as follows:

- For this mission, increasing the engine thrust level from 75,000 to 100,000 lb reduces initial vehicle weight by 6 to 10%, as does increasing the thrust from 100,000 to 200,000 lb. The weight difference between the 75,000-and 200,000-lb thrust engines is 16 and 13% for circular and elliptic orbits, respectively.

- The number of engines required is reduced by two for each increment of thrust level in the circular orbit case and by one for the elliptic orbit case.

- The weight reduction resulting from the use of elliptic orbits is significant, amounting to 28 to 37%. The largest percentage reduction is in the NNSA case.

- Aftercooling and reuse of the nuclear engine leads to a weight reduction of only 6% at the most, and in one case ($NN_{NI}NA$) results in a weight increase. The complete data presented in Volume II also show that engine firing times are longer with aftercooling, exceeding the 2700 sec limit for circular orbit cases.

- On the basis of the "in-line" constraint adopted for the study, the total number of modules required is higher by one or two when the smaller nuclear engines are used or when circular orbits are used.

Table IV-3. Mars Orbital Capture Mission, 1984 IIB Opposition Class

Thrust (lb)	75,000		100,000								200,000	
Vehicle Mode	NNNA		NNNA		NN_JNA		$NN_{NJ}NA$		NNSA		NNNA	
Mars Parking Orbit	C	E	C	E	C	E	C	E	C	E	C	E
Vehicle Wt (10^6 lb)	1.91	1.32	1.71	1.23	1.61	1.61	1.73	1.17	2.60	1.63	1.60	1.15
Configuration	4-2-1	3-1-1	3-1-*1	2-1-1	2*-1-1	2-1-1	2*-1-1	2-1-1	4-2-0	2-*1-0	**1-*1-1	1-*1-1
Number Modules	8	6	7	5	7	5	6	4	8	7	7	5

*Tier 2A required
**Tiers 2 and 3 required
C Circular parking orbit
E Elliptic parking orbit

LAUNCH WINDOW ANALYSIS

The objective of this task was to determine the effect on initial vehicle weight in Earth orbit of providing launch windows for the depart Earth and depart Mars maneuvers. Since the launch window chosen was 20 days at both planets, this means that the vehicle should be sized to carry enough propellant to depart Earth on any day during a 20-day period and, regardless of the actual depart date, to depart Mars on any of the 20 days following the end of the nominal 30-day stopover period.

A complex analysis is involved for providing launch windows at Earth and Mars for round trip interplanetary missions launched from Earth orbit. The parking orbits at Earth and at Mars precess during the launch window period, requiring either a plane change or a nonoptimum trajectory for the interplanetary leg affected. Since it is not known in advance which day during the Earth launch window will be the actual launch date, the vehicle must be sized for the most unfavorable date, and since the date of the return launch from Mars is also unknown, the entire vehicle must be sized for the most unfavorable date in the Mars launch window.

Determining the minimum initial vehicle weights with allowance for launch windows primarily involves selection of parking orbits at both planets and of interplanetary trajectories. Plane changes can be minimized by selecting nonoptimum interplanetary trajectories, or the optimum interplanetary trajectory can always be followed if there is no limitation on plane changes. A tradeoff procedure is obviously required. At least the following three approaches could be applied:

- For each Earth or Mars depart date, find the two-way trajectories that result in the best possible combination of ΔV's without regard to plane changes, for each date in the launch window. Then choose the parking orbits at both planets so that the plane change ΔV's plus the departure ΔV's will be minimized over the whole window.

- Choose the parking orbit so that the vehicle can depart on the optimum trajectory on one day of the launch window. Then on all other dates in the window, depart in the plane of the parking orbit even though the resulting interplanetary trajectories are nonoptimum.

- Combine the above two approaches, choosing the combination of interplanetary trajectories and parking orbits that yields the lowest ΔV including plane change requirements.

The third of these approaches would yield the lowest weight vehicle, but would require development of a new computer program and exceed the scope of the contract. The first approach was selected on the basis of a preliminary analysis that indicated more promise for this technique. A series of manual analyses and computerized steps was used to arrive at the minimum vehicle weight configuration, as described in detail in Volume II.

The launch window analysis was carried out for a representative lander mission for each of three trajectory classes, i.e., a 1984 opposition class mission, a 1986 outbound Venus swingby mission, and a 1988 inbound Venus swingby mission. In all cases the NNNA vehicle configuration was used, with the 100,000-lb thrust engine and the same firing time and launch azimuth constraints as in the previous analyses. The following guidelines were applied:

- A fixed, 20-day launch window was provided at Earth and at Mars, with the Mars launch window starting at the end of the 30-day stopover period.
- A single maneuver was assumed for injection into the interplanetary trajectory at each planet. No staging or dual burn possibilities were investigated.
- Circular parking orbits were assumed at both planets.
- Whenever a vehicle stage contained more propellant than that which would be subsequently required (due to an early launch from Earth in the launch window), the excess was jettisoned as appropriate, either in the parking orbit or interplanetary orbit. No investigation was made of the alternative possibility of transferring the excess propellant to another stage that might require more propellant because of the early launch.
- All propulsion and propellant modules were sized for the amount of propellant required for the minimum-weight vehicle; no attempt was made to limit the modules of a given vehicle to a specified number of equal sized modules.

- The minimum number of nuclear engines per stage was to be used, within the given constraints.

The results of the launch window analysis for the three representative missions are given in Table IV-4, which includes the comparable results for an "optimum" (i.e., no launch window) mission. Detailed results of the various tradeoff analyses leading to these results are given in Volume II.

The largest weight penalty for providing a 20-day launch window occurs for the 1984 opposition class mission, where the weight increase is approximately 59%. One additional engine is required for the Earth depart stage and five additional propellant modules are needed. Actually for the 4-1-1 configuration, the maximum firing time for the depart Earth engines exceeded the 2700-sec limitation by 240 sec. One additional engine and a weight penalty of 30,000 to 40,000 lb would be required to reduce the firing time below the 2700 sec limit.

The smallest weight penalty, approximately 27% or a half million pounds, was found for the 1986 Venus outbound swingby mission. Two additional propellant modules were required, and two additional engines, although the use of a single engine for the arrive Mars stage would have exceeded the firing time constraint by only 100 to 200 sec.

A 34% weight penalty was found for providing launch windows for the 1988 inbound swingby mission, with two additional engines but the same number of modules.

Although these results were strongly influenced by the given assumptions and constraints, it is clear that the provision of launch windows imposes significant weight penalties. It appears desirable to investigate the effects of launch windows for a greater range of missions and the possibilities of reducing the penalties by one or more of the following techniques:

- Dual or multiple impulse injection and staging for the Earth injection stages, combined with elliptic Earth parking orbits

- Propellant transfer between stages

- Combined plane change and nonoptimum interplanetary trajectories (the third approach described above)

Table IV-4. Launch Window Analysis Results

Mission	Optimum			20-Day Launch Windows		
	Vehicle Weight (10^6 lb)	Configuration	Number Modules	Vehicle Weight (10^6 lb)	Configuration	Number Modules
Opposition Class 1984 IB	1.830	3-1-1	6	2.913	4*-1*-1*	11
Outbound Venus Swingby 1986 3A	1.936	3*-1*-1	8	2.453	4*-2-1	10
Inbound Venus Swingby 1988 II3	1.545	2*-1-1	7	2.066	4-1-1	7

*Tier 2A required

The effects of providing launch windows when elliptic parking orbits are used at Mars should also be investigated. In view of the large weight penalties associated with launch windows, a separate investigation should be made to bracket closely the range of Earth and Mars launch window durations that are necessary from an operational standpoint.

V. FUTURE RESEARCH AND ADVANCED TECHNOLOGY

The results of this study indicate the desirability of certain additional technology development and research that would contribute to more detailed and realistic planning of manned interplanetary missions. These areas of research and development are discussed in Volume II and can be summarized briefly as follows:

- Launch Windows. The reasons for further investigation of launch windows were given in the preceding section.

- Elliptic Planetary Parking Orbits. The limited analyses performed under the present study suggested that important weight savings are possible through the use of elliptic parking orbits at the target planet. The requirements imposed by such orbits (e.g., guidance and control accuracy, operational complexity) should be investigated, as should additional techniques for optimizing the characteristics of the ellipse.

- Earth Launch Systems and Operations. The various phases of the present study have been based on the present payload capability of the Saturn vehicle, which in turn affect the number of modules that must be orbited and the parameters of the Earth parking orbits. It would be more realistic to project the capability of the Saturn vehicle (or possibly other vehicles) to the time period being considered for manned interplanetary missions. Launch azimuth constraints for the period should also be examined.

- Vehicle Configuration and Operation. In the present analyses, each propellant tank was sized to fit the impulse requirements, within the maximum size constraint. This leads to a large number of different size tanks, which would probably be impractical from the design and fabrication standpoint. A study should be made of the effects of adopting a standard set of tank sizes. Likewise, two recently proposed techniques for the use of standard stage sizes should be studied: these are the nonintegral burn and the propellant transfer techniques (References 8 and 9).

- Aerodynamic Braking at Earth. Results of the present study show that limiting the Earth arrival velocity to 15 km/sec can impose severe weight penalties, especially during unfavorable opportunities. Technologies for aerodynamic braking at velocities up to 20 km/sec should be investigated.

- Engine Firing Time. The limitation on nuclear engine firing time imposes weight and complexity penalties because of the increase in the number of engines required to provide a given total impulse; this is especially severe where lower thrust engines are used. Theoretical and experimental effort should be devoted to increasing reliable engine firing times to 1 or 2 hours more.

VI. CONCLUSIONS

The broad conclusions that appear from the results given in the preceding sections are summarized below. More detailed discussion of them will be found in Volume II.

- The minimum weight vehicles for the 1980 to 1993 period, within the given launch azimuth constraints, are NNNA vehicles using the 200,000-lb thrust engine and weighing from 1.53 to 2.51M lb.

- Venus swingby trajectories yield the lowest weight vehicles in all years except 1984 and 1990; in the former year the opposition class trajectory yields a lower weight, and in the latter year the Venus swingby mission violates the launch azimuth constraint and the opposition class is the necessary alternative.

- If aerodynamic braking at Earth is limited to a maximum of 15 km/sec, initial vehicle weight is increased by a factor of 1.5 to 2 over that for the case of unlimited aerodynamic braking.

- Orbital capture missions to Mars or Venus can be performed with initial vehicle weights of 1.44 to 1.65M lb, with a weight saving over lander missions (to Mars) of 170,000 to 350,000 lb.

- The use of smaller nuclear engines increases initial vehicle weight and the number of engines required, and the smallest engine considered (75,000-lb thrust) requires more weight than does the intermediate (100,000-lb thrust) engine. Differences are in the range of 3 to 16%.

- Nuclear engine aftercooling yields a weight reduction of, at most, 6% for the mission analyzed, and for some vehicle configurations, it actually increases total weight.

- The use of elliptic planetary parking orbits leads to significant weight reductions, in the range of 28 to 37% for the mission analyzed for all vehicle configurations considered.

- Provision of 20-day launch windows at Earth and at Mars imposes a large weight penalty, ranging from 27 to 59% for the three Mars lander missions analyzed.

VII. REFERENCES

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